BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: The Spacecraft Thermal Environment and Its Implications to CMM Design
Case 730

DATE: June 13, 1968

FROM: R. Gorman

ABSTRACT

The spacecraft radiative thermal environment is examined for the spectrum of all missions ranging beyond 0.25 A.U. from the sun. The environment is found to be approximately common to all spacecraft missions. From the environment, a power limitation on the Common Mission Module (CMM) is derived and shown to be applicable to a variety of spacecraft power systems. The suitability of a CMM for a variety of mission environments and power conversion systems reinforces the feasibility and desirability of the CMM concept for long term spacecraft development.

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MEMORANDUM FOR FILE

INTRODUCTION

A knowledge of the radiative thermal environment at various locations in the solar system is necessary to design a thermal radiator system capable of operating effectively there. The aim of this paper is to examine the radiative thermal environment between 0.25 A.U. and 2.5 A.U., including planet orbits, to see if a single spacecraft thermal design can operate efficiently throughout the entire range. The vehicle is referred to as the Common Mission Module (CMM). A secondary objective is to see what modifications are required, if any, to make a single thermal design accomodate the entire mission spectrum.

THE THERMAL ENVIRONMENT

There are three principal sources of thermal radiation to a spacecraft. These are:

1. solar radiation
2. reflected solar radiation from nearby planets, moons, rings, asteroids, and other spacecraft
3. infrared radiation emitted by warm bodies nearby, such as planets, moons, rings, asteroids and other spacecraft.

The heat transferred to a spacecraft from any of these sources is dependent on:

1. The absorptivity and emissivity of the various spacecraft surfaces.
2. The shape of the spacecraft and the source.
3. The orientation of the spacecraft with respect to the source of the radiation.
4. The radiative properties of the source.

A more detailed thermal analysis is given in Appendix 1.

SPACECRAFT THERMAL COATINGS

The absorptivity of a surface is a function of the wavelength distribution of the illumination. For example, the absorptivity of polished aluminum is ~ 0.25 for sunlight but only ~ 0.03 for infrared radiation. The thermal emissivity of a surface is about the same as the absorptivity for infrared radiation; thus, a surface can radiate heat effectively while not absorbing much sunlight. The absorptivity to solar radiation is called α_s and the infrared emissivity is ε.

In this analysis, two existing surface coatings are considered; both from Lockheed Missiles and Space Company. The first is white silicate paint ($\alpha_s = 0.18$, $\varepsilon = 0.86$) the second is optical solar reflector (OSR) material ($\alpha_s = 0.05$, $\varepsilon = 0.77$).

APPLICATION TO CMM

The CMM consists of a 20' diameter pressure vessel suspended within a 22' diameter structural cylinder. The heat rejection radiators are integral to the structural cylinder. The pressure vessel is thermally isolated from the radiators by high performance insulation and low conductivity load supports (as in the design of cryogenic propellant tanks for space vehicles). Thus, the radiator and pressure vessel exchange energy only via the hot and cold fluids in the ECS (see Figure 1 and Appendix 2).

THE MISSION SPECTRUM

The heat rejecting capacity of the Environment Control System (ECS) for the worst cases of a wide spectrum of possible missions was examined. The analysis assumes that using the total available radiator area, the ECS must reject 10 kw at a radiator temperature of 530°R, or that the Brayton power system must reject 30 kw at 650°R. These common capacity levels we have used to determine if the spacecraft could operate in the various environments. These capacity levels were selected from preliminary analysis. The missions are:

1. Interplanetary trajectories ranging from 0.25 A.U. - out
2. Earth orbit
3. Mars orbit
4. Venus orbit
5. lunar orbit
6. lunar surface

The radiative properties of the planets are taken from Reference 1.

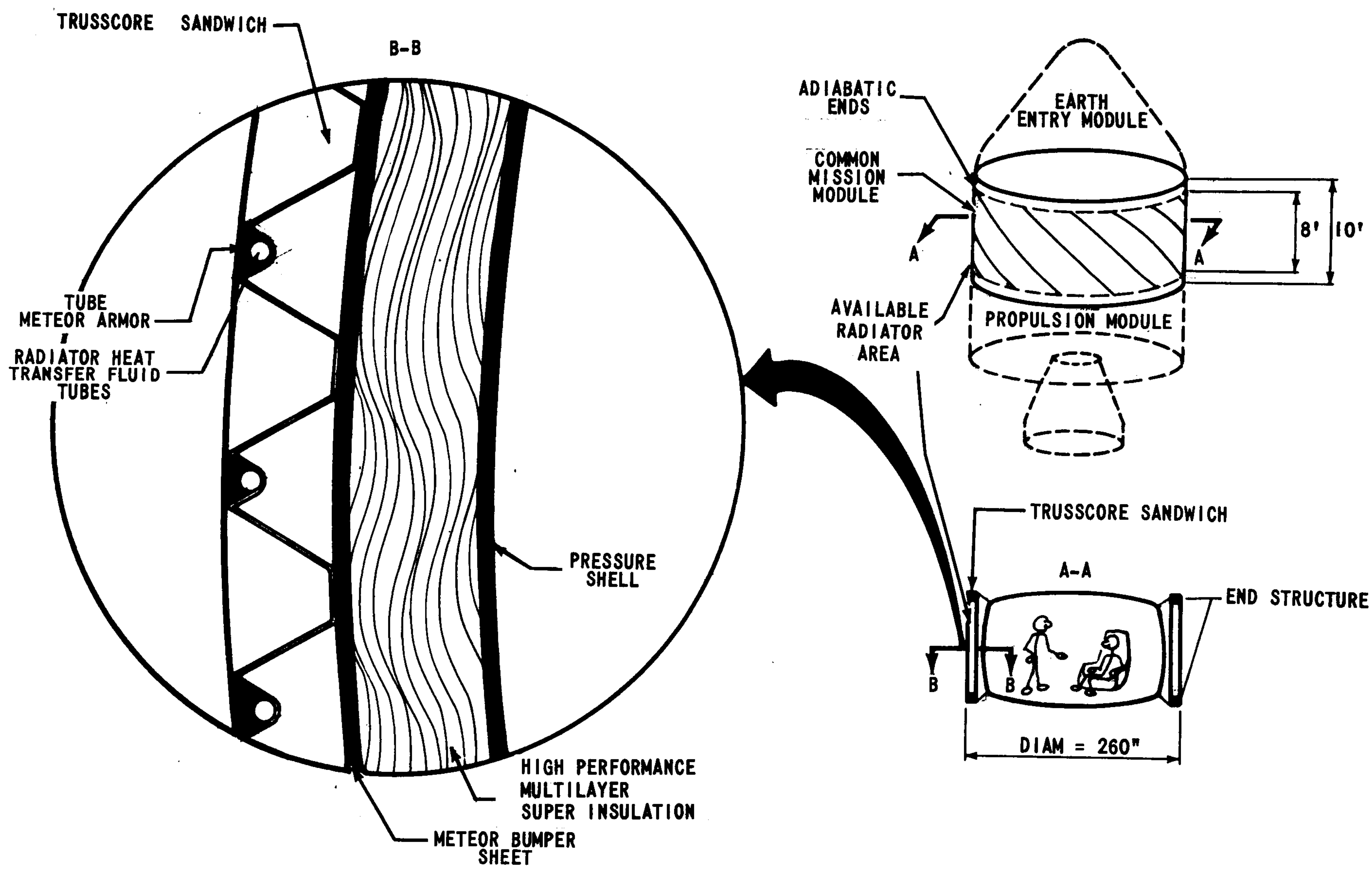

FIGURE 1 - RADIATOR CONFIGURATION

INTERPLANETARY SPACE

The worst attitude for the spacecraft in interplanetary space (only solar radiation heating) is with the spacecraft broadside to the sun. The effects of solar distance and surface coating on heat rejection capacity for a CMM in this attitude is shown in Figure 2.

Beyond 2 A.U., the solar heat load becomes negligible for both the white paint and the solar reflector coating. The solar reflector coating permits operation to within 0.37 A.U. while the white paint will only allow operation to 0.70 A.U. The spacecraft would then be required to orient itself end-on to the sun, so that the radiator area exposed to the solar heat load is reduced. The capacity in the end-on attitude is approximately equal to the capacity beyond 2.0 A.U. If the broadside attitude must be maintained for experimental requirements, the power radiating capacity will be decreased until only the shaded side of the radiator can be used - the tubes on the hot side being shut off and allowed to reach a high temperature thermal equilibrium. The radiating capacity is then approximately one-half the value which can be radiated at large distances (beyond 2.0 A.U.).

ORBITAL MISSIONS

In orbital missions, the attitude that yields maximum thermal radiation load depends on the albedo and solar constant of the body around which the spacecraft is orbiting. The thermal input was determined for various attitudes and orbits (equatorial or polar, perpendicular or parallel to the sun line). The radiator capacity in orbit, about earth, Mars, Venus, and the moon, is shown in Figures 3-6 for the limiting conditions.

Disregarding lunar and Venusian orbits for the moment, the variations throughout an orbit are relatively small. OSR reduces this variation to a negligible quantity for Mars and earth and to about 10% of the maximum rejection capacity for Venus. The lunar orbit case is obviously more serious since the ECS capability is negative in the subsolar region. Generally, the lunar problem is due to an albedo of 0.07 which produces a strong source with an infrared spectrum (See Appendix 1). Although the ECS radiator capacity is negative around the subsolar point, the average heat dissipation capacity over the whole orbit is still quite high. In order to match the heat load over the orbit with the radiator capacity, some method of thermal storage must be employed. If the fluid loop to the radiator is diverted to a heat exchanger which heats stored water during the hottest part of the orbit, the excess heat can then be radiated when the spacecraft is in a cooler segment of the orbit.

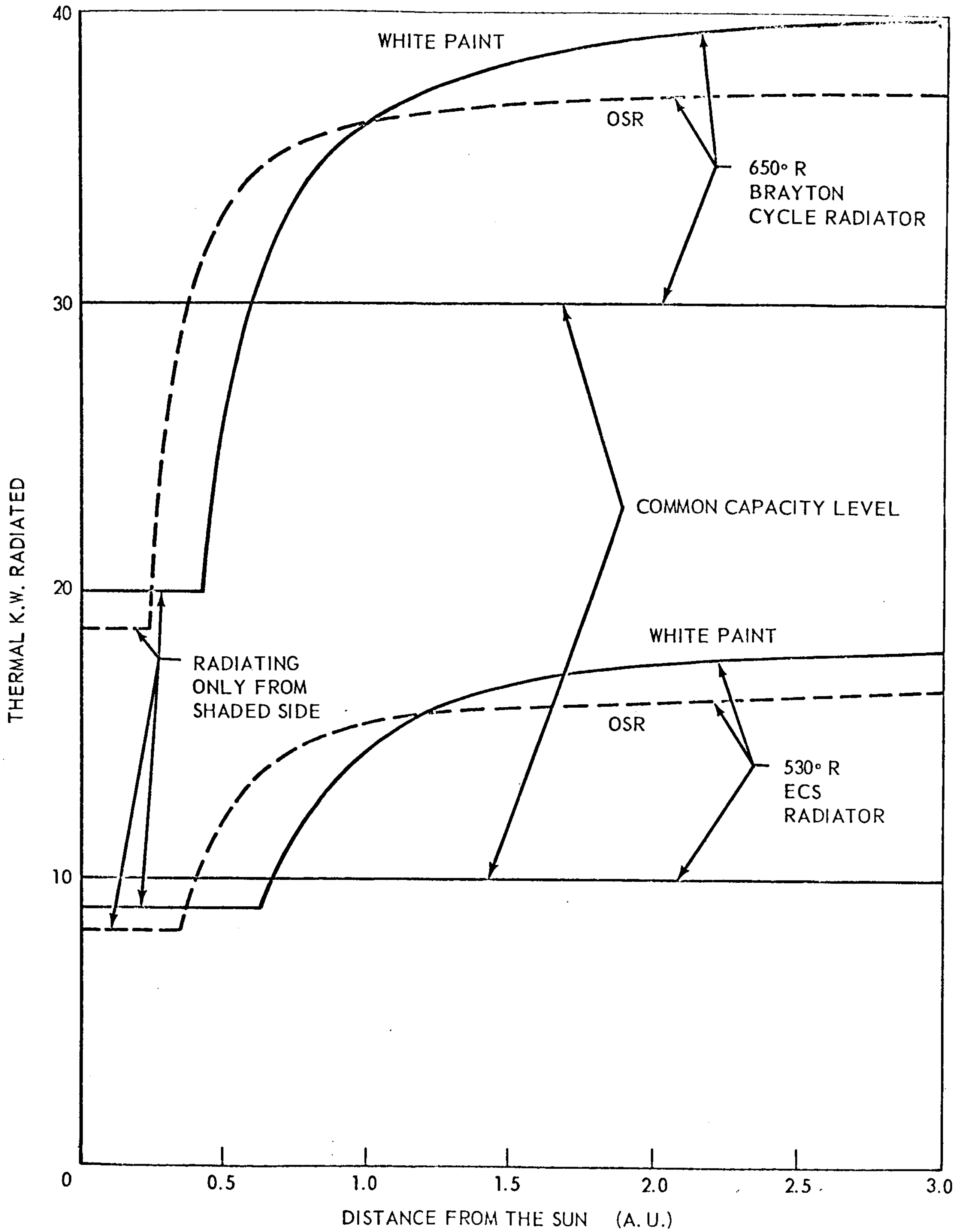

FIGURE 2–INTERPLANETARY RADIATOR CAPACITY

EARTH ORBIT

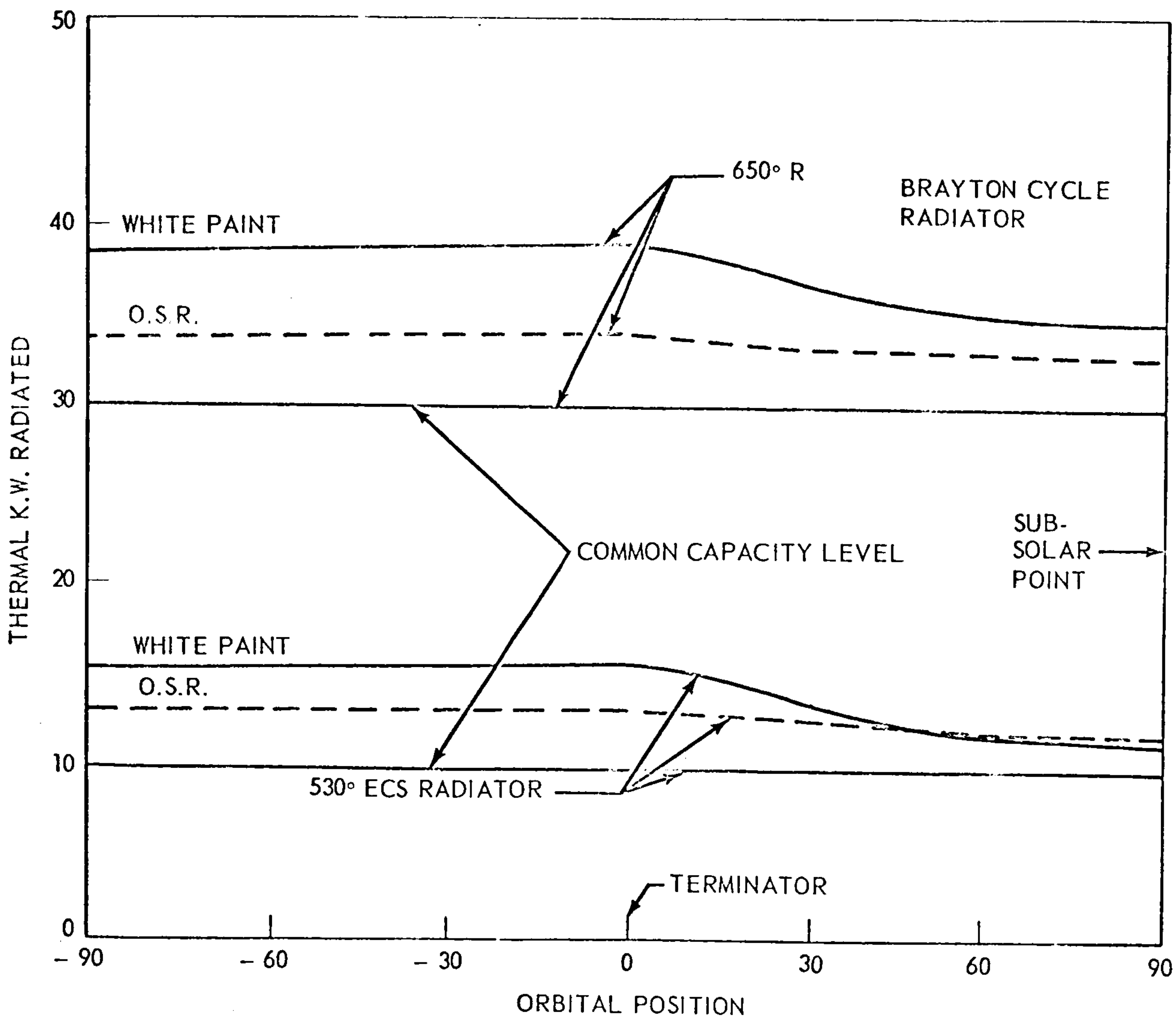

FIGURE 3–HEAT RADIATION CAPACITY IN 100N.MI. ALTITUDE EARTH ORBIT IN THE PLANE OF THE ECLIPTIC BROADSIDE TO THE GROUND

MARS ORBIT

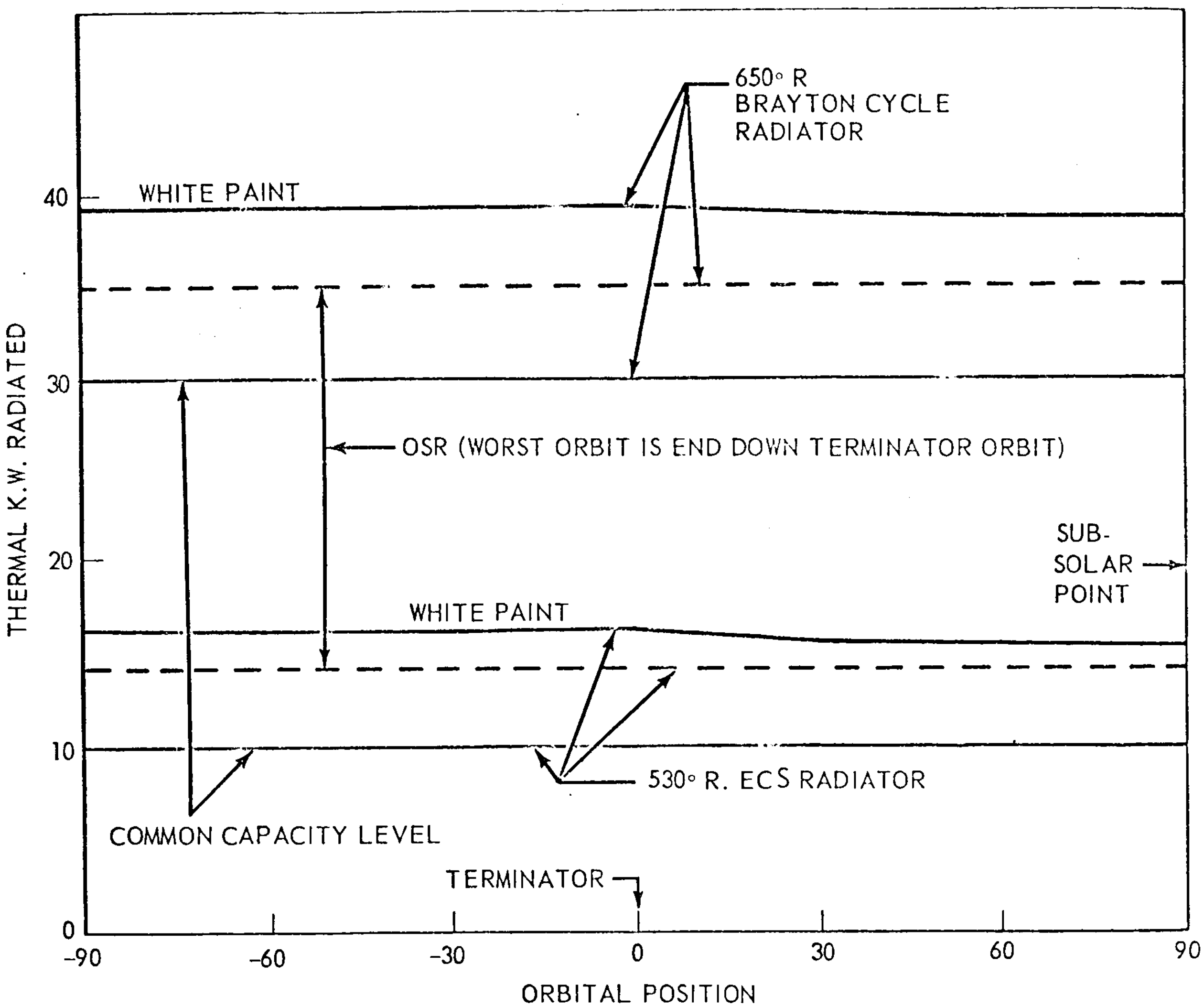

FIGURE 4--HEAT RADIATION CAPACITY IN 53.5 N.M. ALTITUDE
MARS ORBIT IN THE PLANE OF MARTIAN ORBIT
ATTITUDE BROADSIDE TO PLANET SURFACE

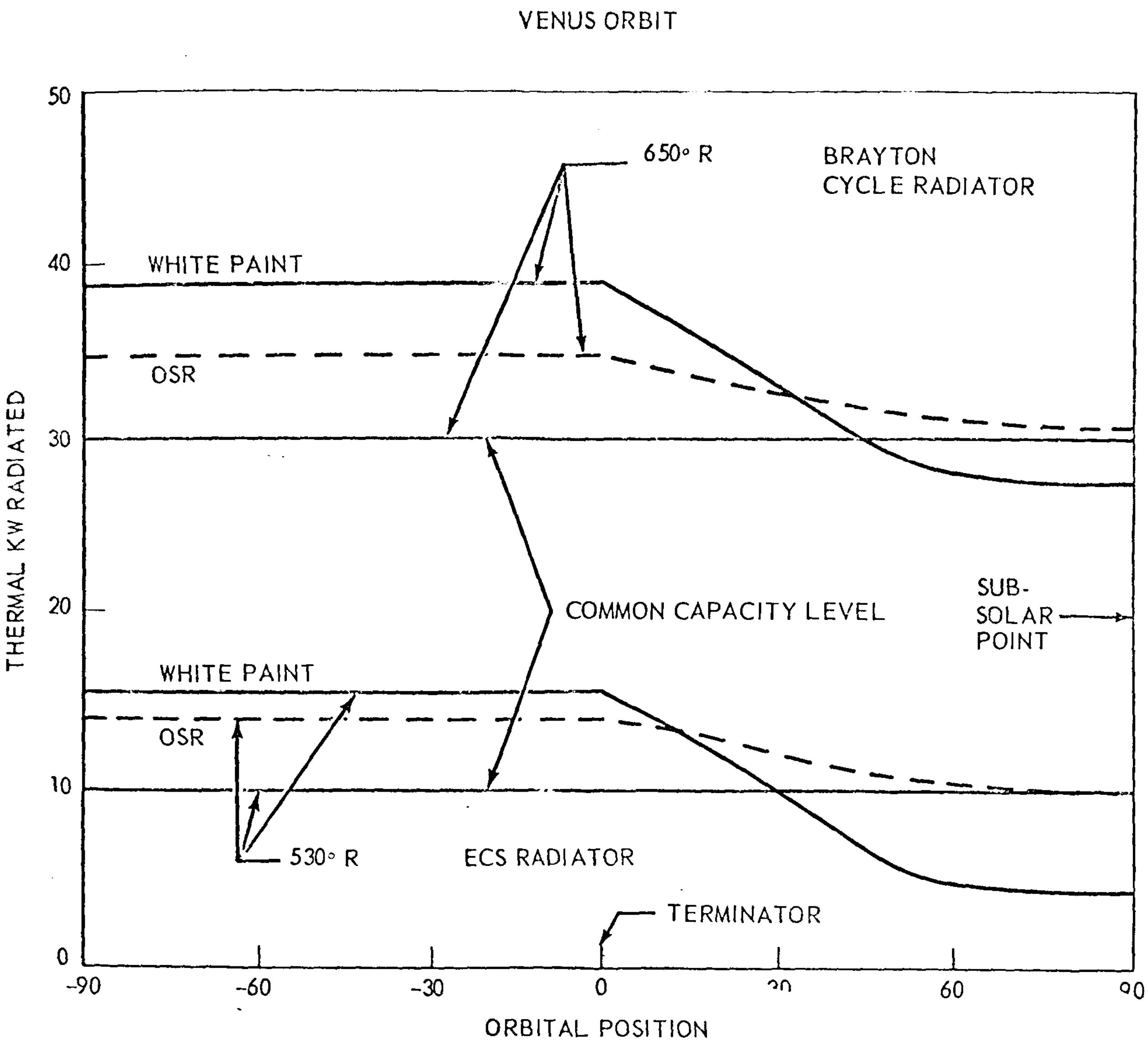

FIGURE 5–HEAT RADIATION CAPACITY IN 95.6 N.M. ALTITUDE
VENUS ORBIT IN THE PLANE OF VENUS ORBIT
BROADSIDE TO SURFACE

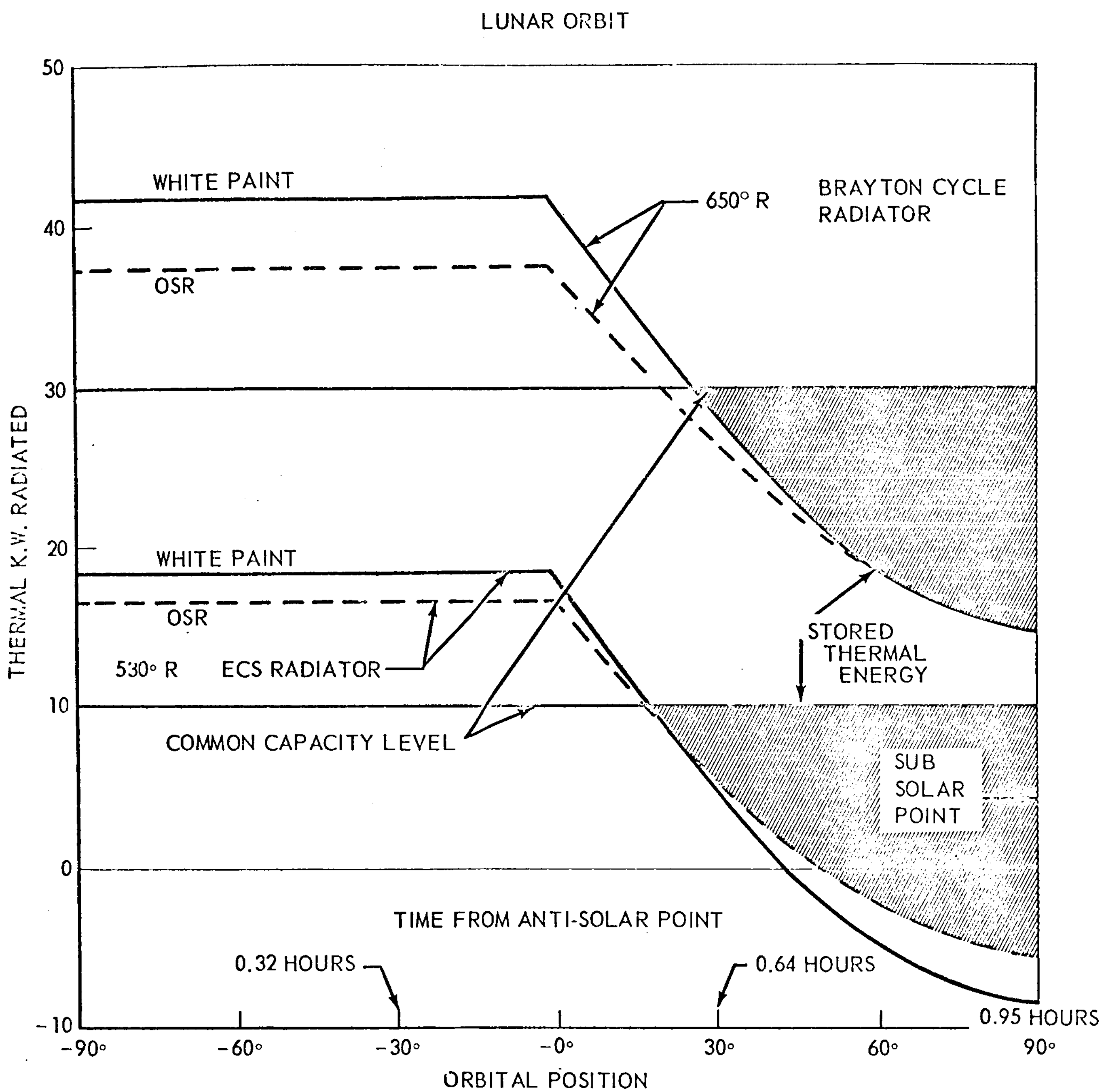

FIGURE 6– HEAT RADIATION CAPACITY FOR 27.5 N.M. ALTITUDE CIRCULAR LUNAR ORBIT IN PLANE OF THE ECLIPTIC BROADSIDE TO PLANET PERIOD = 1.89 HOURS

The total energy stored per orbit is about 10 kwh. This can be stored as the thermal swing of 500 lbs of water through 70°F. 500 lbs of water is less than the stored water necessary for reserve supply for even a small crew, and the weight of the required extra tankage, pump, valves, and heat exchangers will be about 50 lbs or less.

LUNAR SURFACE SHELTER

On the lunar surface, the worst thermal radiation environment is at lunar noon, at the lunar equator. When the sun is between ± 45° from zenith, the surface temperature of the bare lunar soil becomes very high. The period of high thermal radiation lasts for about seven earth days, so a steady state rather than a transient solution is needed.

In order to keep the radiator configuration the same, a reflective mat could be deployed on the lunar surface around the spacecraft. Figure 7 shows the configuration of the spacecraft and the mat, and the variation of radiator capacity with mat radius and weight. Such a mat, with a high solar reflectivity converts the nearby lunar surface from an infrared source to a source with a solar spectrum. A mat about 300' in diameter and weighing about 250 lbs would be needed.

PRELIMINARY CONCLUSIONS

Some preliminary conclusions, assuming an ECS load of 10 kw or a 650°R Brayton Cycle radiator load of 30 kw, are:

1. A common thermal design can be used for potential missions. Minor modifications are required for:

   - lunar surface
   - low lunar orbit
   - low Venus orbit

2. These modifications are:

   a) On the lunar surface, a solar reflective mat must be deployed and OSR material must be used on the CMM surface.

   b) In a low lunar orbit, a transient technique such as heating and cooling storage water is required to handle hot portions of the orbit.

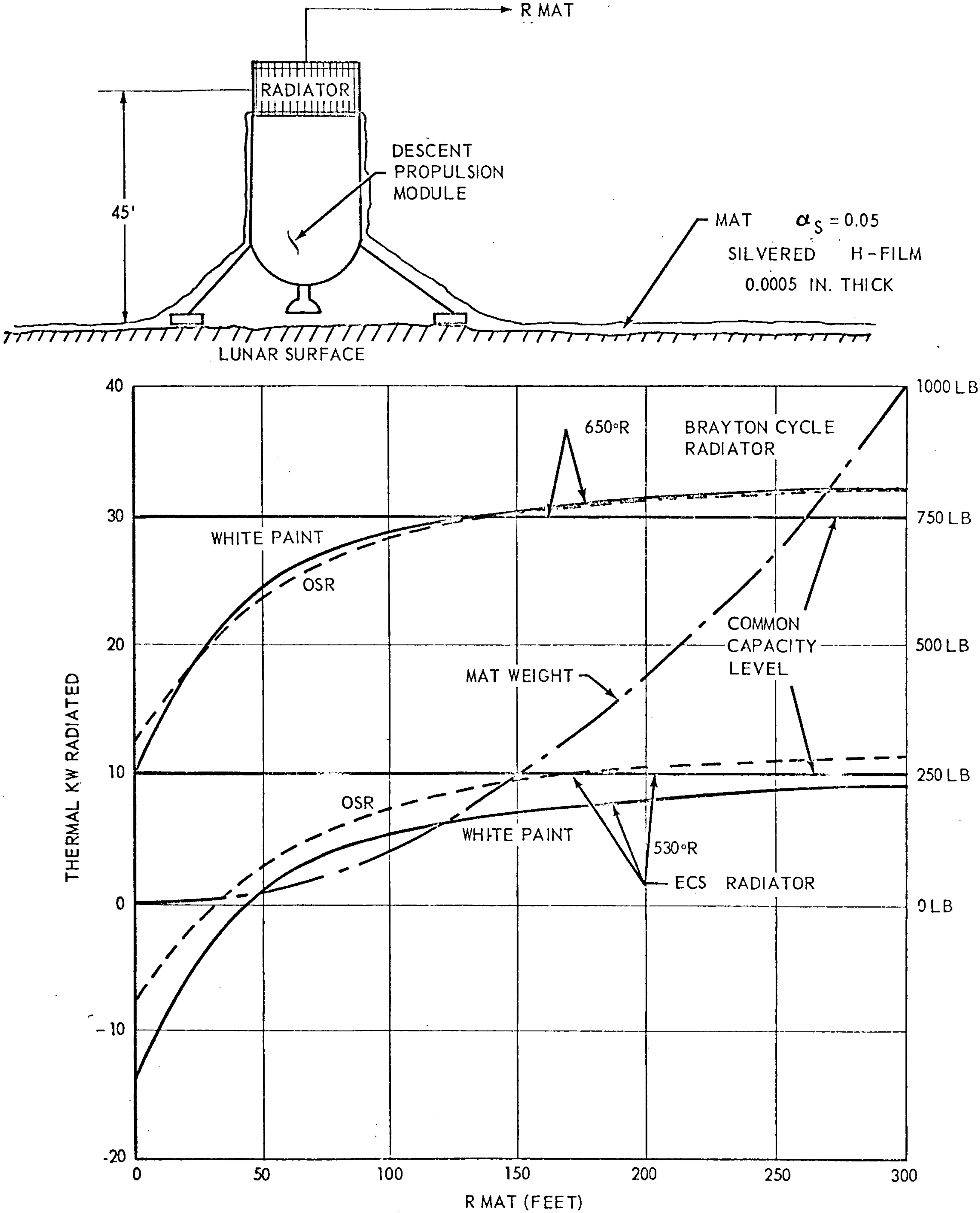

FIGURE 7–LUNAR NOON

c) For worst Venus orbit OSR material or a thermal storage as in 3 above, must be used.

d) Close in to the sun, (less than 0.37 A.U.) the spacecraft cannot be oriented to the worst possible configuration.

COMMON MISSION MODULE DESIGN IMPLICATIONS

The geometry and environment of a CMM define the maximum radiator capacity for either ECS or space power system use. However, the CMM carries both an ECS and a power system so that the available radiator area must be shared between the two systems. All long mission space power systems other than solar cell arrays require heat rejection. These are the thermodynamic power systems (Brayton cycle, Rankine cycle, Stirling cycle, Thermionic effect, Thermoelectric effect) and all require heat rejection at the lowest cycle temperature.

The amount of heat which must be rejected by the space power system is determined by the desired power output and the overall efficiency of the power system. The radiator area required per kw of electrical output is shown in Figure 8 for space power systems of varying overall efficiency and radiator temperature. The effective sink temperature for this calculation represents one of the worst cases, namely Venus orbit.

Most of the heat generated in the cabin comes from the electric power consumed there. The only parts of the electrical input not turned directly into heat are:

1. the power which is converted to radio waves by the communications systems;

2. the power consumed in the electrolysis of water for oxygen regeneration;

3. the power used to compress the gases used in the spacesuit life support systems or reaction control systems; and

4. the energy used by experiment packages and mechanisms which are not in the cabin.

Of these, the only significant energy sink is the electrolysis of water. Even this energy ultimately finds its way back into the cabin as heat because the crew breathes the resulting oxygen, produces H_2O and CO_2 and releases the heat of reaction as metabolic heat.

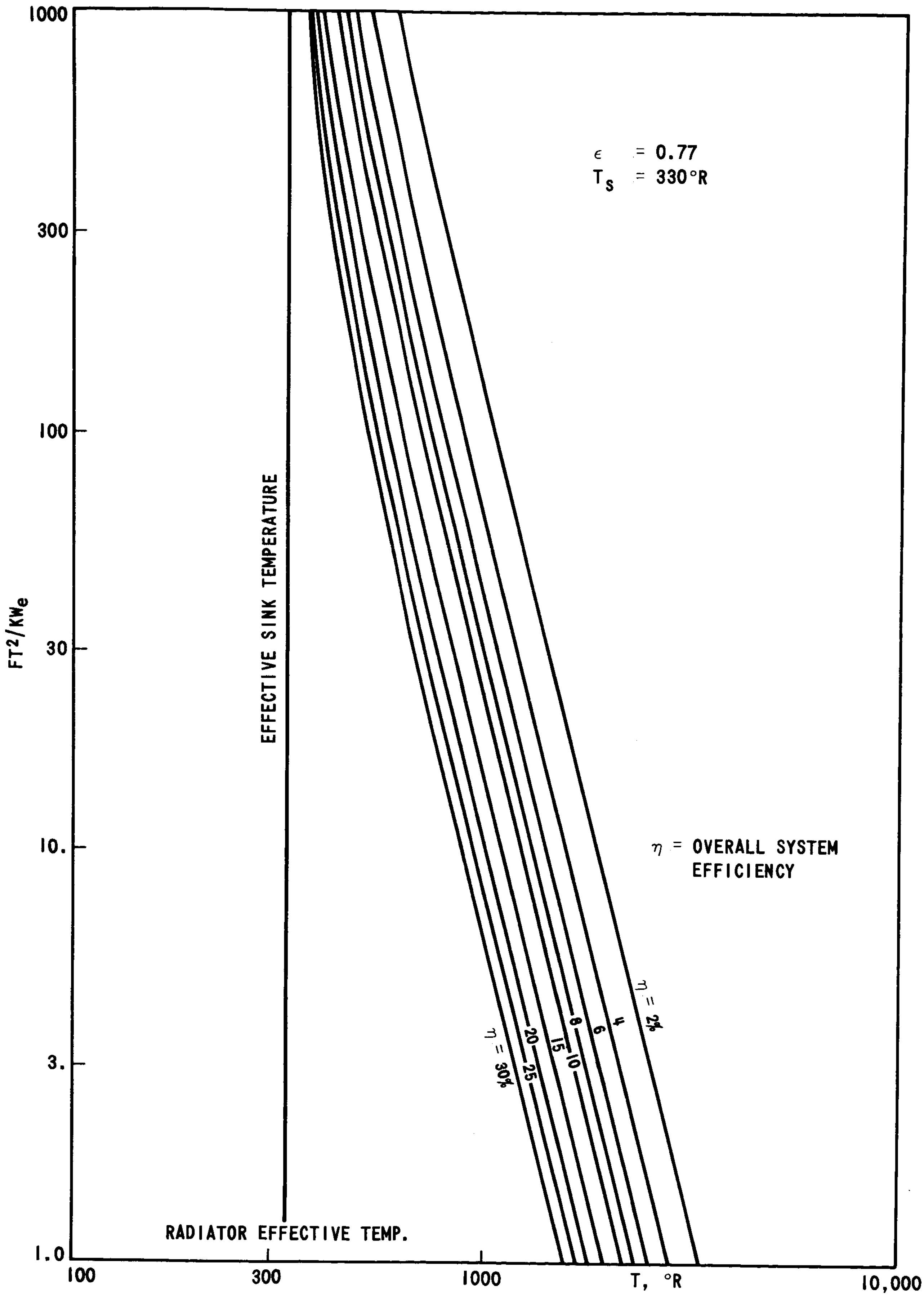

FIGURE 8 - THERMODYNAMIC POWER SYSTEM RADIATOR REQUIREMENTS

The cabin heating supplied by other than the power system may be a significant part of the total load. These other sources include such things as "waste heat" and radioisotope heat sources supplying process heat for the life support system. There may also be heat leaks from the power conversion system lubrication and control system. These cooling loops may require either separate intermediate temperature radiators or may be pumped directly into the ECS cooling loop. In any case, the total ECS cooling load is the output of the electrical power system plus an extra small load due to endogenous cabin heat sources.

AVAILABLE RADIATOR AREA SHARING

There will be a limit on the spacecraft power system power which is imposed by the area available to radiate the cycle rejected heat, and the cabin electrical load and endogenous sources. The less radiator area needed for the space power system, the more area available for rejecting its output as ECS output. There is a maximum power which can be generated for the given area of the spacecraft outer shell. Therefore, the maximum electric power usable on the spacecraft is a function of the achievable power system overall efficiency and its radiator rejection temperature. Figure 9 gives the usable electric power (with and without a 2 kw endogeneous cabin heat load) as a function of the achievable power system overall efficiency and rejection temperature, for a CMM radiator as described in Figure 1. Several alternative space power systems are indicated in Figure 9 so that the power level which can be accommodated for each system can be compared.

For modules of similar configuration (cylinders) the analysis is similar. The capacity of an integral radiator will be proportional to the cylindrical area available. Thus, a 15' diameter module which is 10' high will only be able to radiate 15/22 as much power and its thermal radiation capacity would be limited to 15/22 the capacity of a 22' module.

CONCLUSIONS

The configuration of the CMM assumed can radiate enough heat from its outer wall to accommodate the power system heat rejection and ECS heat rejection for a cabin power input of approximately 5 kwe (3.45 kwe for a 15' d. x 10'h, 7.5 kwe for a 33' d. x 10'h module, 7.5 kwe for a 22' d. x 15' high module). This power level can be achieved by several power systems (Organic Rankine Cycle, Xe-He Brayton Cycle, Cascaded Thermoelectric Cycle). The decision of which power system to use is not therefore dictated by radiator area limitation, but by other factors. The fact

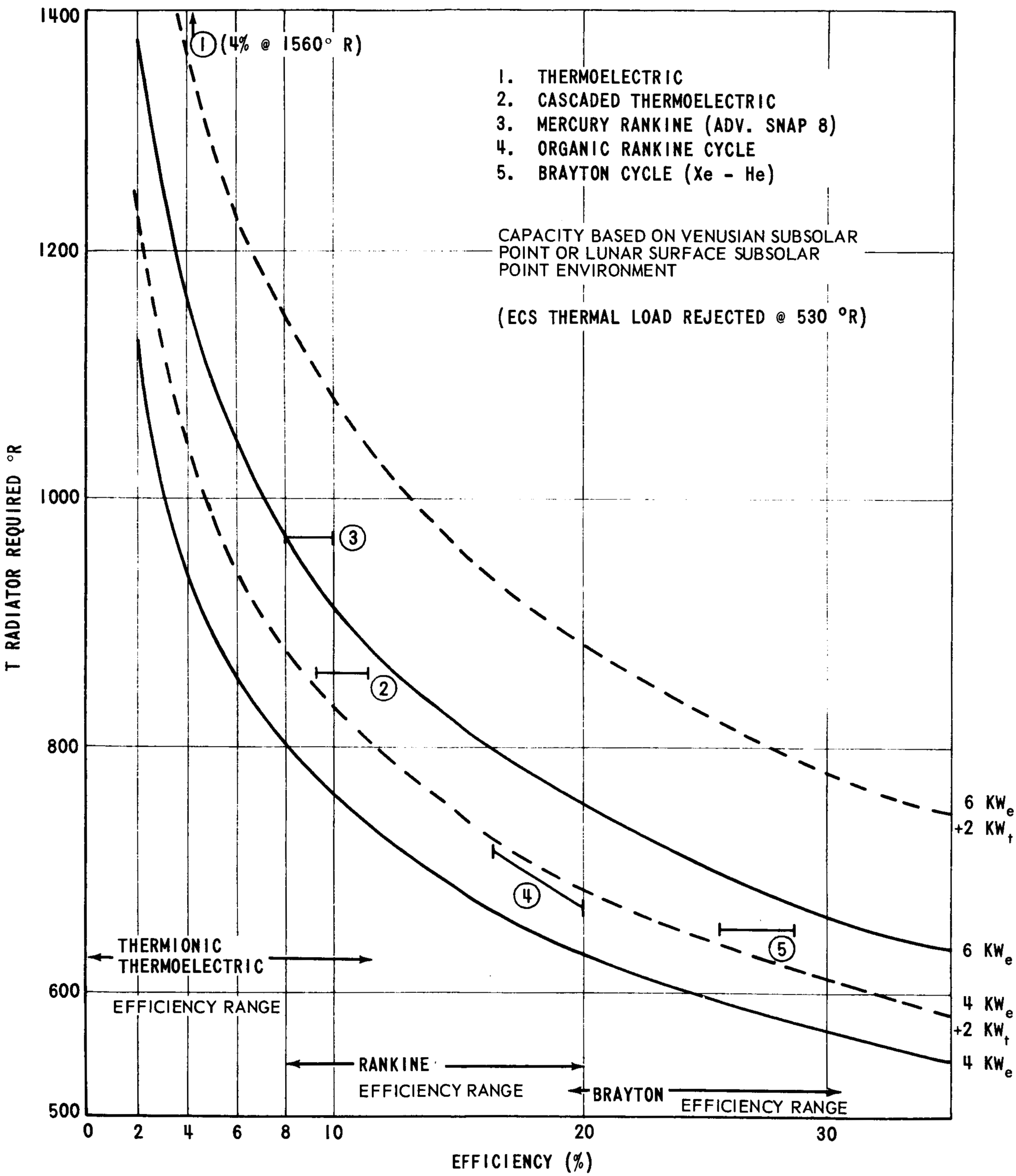

FIGURE 9 - POWER SYSTEM REJECTION TEMPERATURE

that this power level is achievable in all of the proposed CMM environments and with a variety of power systems, reinforces the feasibility and desirability of the CMM concept for long term spacecraft development.

Richard Gorman

1013-RG-pap R. Gorman

Attachments
References
Appendices I and II
Figures 1 to 9

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APPENDIX I

At thermal equilibrium (steady state). (From Ref. 1)

$$\varepsilon\ \sigma T^4 = Q + q_S\ [\alpha_S F_S + A\ \alpha_S F_{SR} + \frac{(1-A)}{4} F_{TR}\ \alpha_{TR}]$$

Where

Q = desired heat radiation from surface

q_S = solar constant at that distance from the sun

$$\frac{442\ B_{TU}}{Ft^2\ hr.} @ 1\ A.\ U.$$

F_S = geometric factors relating shape, orientation of vehicle with position of sun for direct sunlight.

F_{SR} = geometric factor relating shape, orientation of vehicle with planet with respect to reflected solar radiation.

F_{TR} = geometric factor relating shape, orientation of vehicle with planet with respect to infrared emission of planet (due to sunlight absorbed by planet).

A = Albedo of planet -- amount of incident sunlight reflected from it i.e., $(1-\alpha_S)$ of the planet

α_S = Solar absorptivity, i.e., absorption of light of visible wave lengths $\sim 0.4\mu$

α_{TR} = Infrared absorptivity, i.e., absorption of the thermal radiation characteristic of warm bodies $\sim 5-20\mu$

ε = Emissivity of body at temperature of body; usually equals α_{TR}

σ = Stefan-Boltzman radiation constant

$$= 0.17 \times 10^{-8} \frac{BTU}{Ft^2\ hr.\ (°R^4)}$$

This relationship holds everywhere in the solar system except on the moon and in orbit around it. The moon is different because of three factors.

1. The rotation rate is very slow,
2. The Albedo is very low,
3. The surface is made of material with very small conductivity.

This results in a surface that is in local thermal equilibrium so that

$$\left(\frac{1-A}{4}\right)\left(F_{TR}\right)\alpha_{TR} \text{ becomes } \left(1-A\right)\left(F_{SR}\right)\alpha_{TR}$$

because the local thermal emission is proportional to the local incident solar intensity.

APPENDIX II

CMM Configuration

The CMM configuration is shown in Figure 1. The ends are shielded by other components of the mission assemblage (propulsion stages, re-entry module) so that the only available surface for radiation is the cylindrical side wall. The ends of the cylindrical surface are taken up by the coupling sections so that the radiating heat-rejecting area is limited to a cylinder 8 ft. in height and 21.65 ft. in diameter or an area of 544 square ft. A temperature of 530°R is selected as the effective radiator temperature for the environmental control system (ECS) rejected heat. A temperature of 650°R is selected as a representative effective Brayton Cycle Power System radiator temperature.

Two surface coatings are used for the calculations:

1. LMSC white silicate paint $\alpha_S = 0.18$ $\varepsilon = 0.86$
2. OSR (Optical Solar Reflector material)

   $\alpha_S = 0.05$ $\varepsilon = 0.77$ (See Reference 2)

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